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TECHNICAL NOTE

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WIND-TUNNEL CALIBRATIONS OF A COMBINED PITOT-STATIC TUBE, VANE-TYPE FLOW-DIRECTION TRANSMITTER, AND STAGNATION-TEMPERATURE ELEMENT AT MACH NUMBERS FROM 0.60 TO 2.87

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WIND-TUNNEL CALIBRATIONS OF A COMBINED PITOT-STATIC TUBE, VANE-TYPE FLOW-DIRECTION TRANSMITTER, AND STAGNATION-TEMPERATURE ELEMENT AT MACH NUMBERS FROM 0.60 TO 2.87

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SUMMARY

Calibration tests of a combined pitot-static tube, vane-type flow-direction transmitter, and stagnation-temperature element have been conducted over a range of Mach numbers from 0.60 to 2.87 in the Langley 8-foot transonic pressure tunnel and in the Langley Unitary Plan wind tunnel. The results indicate that the variations in static-pressure error due to angle of attack were generally less than 1 percent of the impact pressure for angles up to 15°, whereas the variations due to a sideslip angle of $\pm 10^{\circ}$ ranged from 2 to 4 percent, depending on the Mach number. The effects of angles of attack from -3° to 20° and/or angles of sideslip from -10° to 10° on the measurement of total pressure were less than one-half of 1 percent of the impact pressure. The angle-of-attack vane indicated too high an angle with an error at 15° ranging from 0.5° to 2.4°, depending on the Mach number. The sideslip-vane errors ranged from zero to 1.5° for sideslip angles to $\pm 10^{\circ}$. There were no significant variations in the measured stagnation temperature due to angles of attack from about -3° to 20° and/or angles of sideslip from -10° to 10°.

INTRODUCTION

A combined pitot-static tube, vane-type flow-direction transmitter, and stagnation-temperature element has been designed by the NASA for use on research aircraft. The instrument is designed to mount on a nose boom and combines a pitot-static tube, two free-floating vanes for measuring the angles of attack and sideslip, and a stagnation-temperature element. The unit tested is generally similar to ones reported previously (refs. 1 and 2) except for the addition of the stagnation-temperature element which was positioned opposite the angle-of-attack vane strut to improve the symmetry of the flow field for the angle-of-sideslip vane. Previous calibrations of a similar unit without the temperature element at Mach numbers of 1.61 and 2.01 (ref. 2) had shown sideslip-vane errors as large as 1.4° at zero angle of sideslip.

The purpose of the present investigation was to extend the range of calibration Mach numbers and to assess the effects of the presence of the temperature element. Tests were conducted in the Langley 8-foot transonic pressure tunnel over the Mach number range from 0.60 to 1.20 and in the Langley Unitary Plan wind tunnel over the Mach number range from 1.47 to 2.87. Calibration data are presented to show the effects of Mach number and of angles of attack and sideslip on the measurement L
of total-pressure, static-pressure, and flow direction. 4
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SYMBOLS

M_∞	free-stream Mach number
p	static pressure measured by tube
p_t'	total pressure measured by tube at zero angle
p_t''	total pressure measured by tube at angles other than zero
p_∞	free-stream static pressure
q_c'	impact pressure, $p_t' - p_\infty$
$\frac{\Delta p_t}{q_c'}$	error in total pressure, $\frac{p_t'' - p_t'}{p_t' - p_\infty}$
$\frac{\Delta p}{q_c'}$	error in static pressure, $\frac{p - p_\infty}{p_t' - p_\infty}$
α	angle of attack, nose up positive
$\Delta\alpha$	angle-of-attack vane correction, $\alpha_{true} - \alpha_{meas}$
β	angle of sideslip, nose to left positive

$\Delta\beta$ angle-of-sideslip vane correction, $\beta_{true} - \beta_{meas}$

Subscript:

meas measured

APPARATUS AND TESTS

The instrument calibrated in this investigation (fig. 1) combines a pitot-static tube for sensing total and static pressures, free-swiveling vanes for sensing angles of attack and sideslip, and a stagnation-temperature element. This instrument was similar to the one previously calibrated (ref. 2) in the Langley 4- by 4-foot supersonic pressure tunnel. The external differences are the addition of the temperature element and a small increase in the diameter of the enlarged afterbody.

Details of the pitot-static tube are shown in figure 2. In order to minimize the pressure errors due to positive angle of attack, an asymmetrical static-pressure orifice arrangement and a slanted total-head opening were used. There were two identical groups of static-pressure orifices, forward and rearward, and pressure from each group was measured independently.

The flow-angle measuring components consist of free-swiveling mass-balanced vanes mounted on the enlarged afterbody as shown in figure 1. The angular position of each vane with respect to the body of the instrument was measured by means of a synchro-transmitter attached to the inboard end of the vane shaft and recorded photographically by a recorder using appropriate synchro-receivers.

The stagnation-temperature element is mounted on the afterbody opposite the angle-of-attack vane strut. This location was selected to improve the symmetry of the flow field for the angle-of-sideslip vane and reduce the errors shown in reference 2. The resistance-type stagnation-temperature element is similar to the one described in reference 3.

Transonic Wind-Tunnel Tests

Tests at Mach numbers from 0.60 to 1.20 were conducted in the Langley 8-foot transonic pressure tunnel. The instrument was mounted by the use of an adapter on the regular model support sting as shown in figure 3. In the orientation shown, the angular travel of the sting provided a range of angles of attack from -3° to 20°. Combined angles of attack and sideslip were obtained by the use of an angled coupling between the adapter and the sting to provide a fixed angle of sideslip. In order to cover an angle-of-sideslip range at zero angle of attack, the instrument was rotated 90° on the adapter. The test angles were set by a remote indicating inclinometer in the adapter to eliminate any effects of sting bending due to airload. Allowance was made in these settings for a downflow of 0.2° in the previously calibrated test section.

The free-stream total pressure was obtained from the total pressure L
measured in the low-speed section upstream of the tunnel contraction 4
cone. The free-stream Mach number and static pressure were determined 2
from simultaneously measured values of the free-stream total pressure 5
and test-chamber static pressure used in conjunction with tunnel calibration curves. Total and static pressures from the instrument were measured with standard liquid manometer tubes with the free-stream total pressure used as the reference.

All tests in this tunnel were made with a free-stream total pressure of 1 atmosphere. The Reynolds number per foot ranged from about 3.14×10^6 at a Mach number of 0.60 to 4.16×10^6 at a Mach number of 1.20.

Supersonic Wind-Tunnel Tests

Tests at Mach numbers from 1.47 to 2.87 were conducted in the low Mach number test section of the Langley Unitary Plan wind tunnel. The instrument was mounted by an adapter to an elevation-angle mechanism which, in turn, was mounted on the regular model support sting. The azimuth angle of the sting was controllable so that various combinations of angles of attack and sideslip could be obtained. Figure 4 shows the instrument installed in the tunnel.

Previous calibrations of the tunnel had indicated an upflow angle in the test section ranging from near zero to about 1.5° depending on the Mach number. In the Mach number range covered, it was assumed that the angle-of-attack vane would indicate correctly near zero angle of attack as found in the tests of reference 2. When the instrument was geometrically zeroed in the tunnel, the stream flow angle indicated by the angle-of-attack vane agreed very closely with the tunnel calibration values. The actual flow angle for the probe in these tests was then taken as the sum of the geometric angle of the probe and the stream flow angle measured at each Mach number.

The free-stream total pressure was obtained from the total pressure measured in the chamber located ahead of the Mach number 1 contraction section. Total and static pressures from the probe were recorded on sensitive mechanical-optical absolute pressure recorders. The free-stream Mach number was determined from the ratio of the measured total pressure (behind the normal shock) to the free-stream total pressure; the free-stream static pressure was then derived as a Mach number function of the free-stream total pressure.

The free-stream total pressures were such that the Reynolds number per foot was 1.91×10^6 at Mach numbers of 1.47 and 2.01 and was 1.79×10^6 at Mach numbers of 2.36 and 2.87. Limited data were also taken at Mach numbers of 1.47 and 2.01 with the Reynolds number per foot increased to 3.82×10^6 (referred to as "doubled Reynolds number" in figures).

PRECISION OF DATA

In the vicinity of the instrument, the maximum deviations in the free-stream Mach number were estimated to be about 0.003 for Mach numbers up to 1 and about 0.01 for Mach numbers above 1.

In the transonic tests, the error in determining $\Delta p/q_c'$ was estimated to be less than 0.004 below a Mach number of 1 and 0.01 above a Mach number of 1. In the supersonic tests, this error was estimated to range from less than 0.01 at a Mach number of 1.47 to 0.005 at a Mach number of 2.87.

The possible error in determining $\Delta p_t/q_c'$ was estimated to be less than 0.003.

It was estimated that the alinement of the probe with the relative airstream in the direction of the principal measurements was known within 0.2°. The possible error in measuring the angularity between the vanes and the probe was estimated to be about 0.2°.

RESULTS AND DISCUSSION

Pitot-Static Tube

Total-pressure error.- The variation of total-pressure error with angles of attack at various angles of sideslip and at several Mach numbers is given in figure 5. Over the angular range covered (α from about -3° to 20° and β from -10° to 10°), there was no appreciable total-pressure error (less than 0.005). This result is in agreement with the tests of the type A-6 tube reported in reference 4. The random errors shown are considered an indication of the accuracy of measurement and the nonuniformity of tunnel flow more than a direct indication of flow-angle effects.

It should be noted that the total-pressure error was determined as the difference between the pressure measured by the pitot tube at a given angle and the pressure measured at zero angle. For the supersonic tests where shock formed ahead of the tube, this test procedure, in
effect, corrected for the total-pressure loss through the shock; thus, L
the total-pressure errors shown for supersonic speeds are a function 4
only of changes in angle and are therefore directly comparable with the 2
subsonic data. 5

Static-pressure error.- The variation of static-pressure error with angle of attack ($\beta = 0^{\circ}$) at several Mach numbers is shown for the forward and rearward set of orifices in figures 6 and 7, respectively. For the test Mach numbers shown (0.60 to 2.87), the variation in static pressure error due to angle of attack was less than 1 percent of q_c' for angles up to 15°. For this range of angle of attack, the unsymmetrical orifice arrangement provides an average pressure that is very near the pressure measured at zero angle of attack. As the angle of attack is increased above 15°, the static-pressure error, in general, becomes increasingly positive; however, at a Mach number of 1.47 this trend is reversed with the error becoming increasingly negative as the angle of attack is increased to about 22°. A schlieren photograph of the configuration at $M_\infty = 1.47$ and $\alpha \approx 22^{\circ}$ (fig. 8(a)) shows a region of separated flow along the upper side of the static tube. This separated region apparently reduced the pressure of the top orifices to the extent that it outweighs the positive pressure increment of the bottom orifices and thus results in an average pressure which is lower than the stream static pressure. For comparison, a schlieren photograph of the same configuration at the same attitude but at $M_\infty = 2.01$ is presented in figure 8(b). This photograph shows a distinct shock pattern on the upper side of the probe rather than the separated region seen at the lower Mach number.

At Mach numbers of 1.47 and 2.01, an increase of Reynolds number per foot from 1.91×10^6 to 3.82×10^6 had very little effect on static-pressure error for angles of attack up to about 10°. At higher angles, the increased Reynolds number tended to make the pressure error more positive.

There appeared to be no significant differences in the pressure errors for the two sets of static orifices; consequently, the remaining data and discussion are given only for the forward set of orifices.

The data of figure 6 are replotted in figure 9 to show the variation of static-pressure error with Mach number at several angles of attack. In the Mach number range from 0.60 to 0.98, a positive pressure error ranging from about 1 to 5 percent of q_c' occurs in the angle-of-attack range covered. This error is believed to be due primarily to the blocking effect of the enlargement of the probe downstream of the static orifices. In the Mach number range from 1.00 to about 1.05, the data are of an erratic nature primarily due to boundary-reflected disturbances. In the supersonic range, the static-pressure error does not exceed one-half of 1 percent of q_c' for angles of attack up to 10°. At an angle of attack of 20°, the error ranges from about -1 to 2 percent of q_c'.

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The variation of static-pressure error with angle of sideslip ($\alpha = 0^\circ$) at several Mach numbers is shown in figure 10. Because of the asymmetrical arrangement of the static orifices, the static-pressure error due to sideslip is much larger than that due to angle of attack and, as expected, is in a negative direction. At $M_\infty = 0.60$ the error varies from about 1 percent at $\beta = 0^\circ$ to -3.3 percent at $\beta = 10^\circ$ whereas at $M_\infty = 2.87$ the error varies from about -0.3 percent at $\beta = 0^\circ$ to -2.0 percent at $\beta = 10^\circ$.

The variation of static-pressure error with angle of sideslip at various angles of attack is given in figure 11 for several Mach numbers. For angles of attack up to 10°, there appears to be relatively little effect of angle of attack on the magnitude of the error due to sideslip. At an angle of attack of about 15°, however, there was a more pronounced effect of angle of attack.

It should be pointed out that the static-pressure errors are given in terms of the impact pressure q_c'. Charts for converting $\Delta p/q_c'$ to $\Delta p/p$ and $\Delta p/q_c'$ to $\Delta M_\infty/M_\infty$ are given in reference 5. Also, it should be noted that the pressure errors shown herein are for constant static-pressure conditions. An evaluation of the interference effects of flow through the orifices when the static pressure changes rapidly is given in reference 6.

Flow-Direction Vanes

Angle-of-attack vane.- The variations of the angle-of-attack vane correction with angle of attack at several Mach numbers and angles of sideslip are given in figure 12. At $\beta = 0^\circ$, the vane indicated too high an angle of attack due to upwash from the probe except at $M_\infty = 2.87$ at $\alpha \approx 22^\circ$. The corrections are roughly linear with a slope ranging from about 0.5° in 15° to 2.4° in 15°, depending on the Mach number; the maximum correction occurred at $M_\infty = 1.20$. In the subsonic range, the combined effect of sideslip on the angle-of-attack vane correction is relatively small in the angular range of the tests. In the transonic range, there is a large effect of positive sideslip throughout the test angle-of-attack range. At $M_\infty = 1.47$ and above, there is little combined effect shown until the angle of attack exceeds about 10°.

The data at $\beta = 0^\circ$ of figure 12 are replotted in figure 13 to show the variation of the angle-of-attack vane correction with Mach number at $\alpha = 0^\circ$, 10°, and 20°. In the transonic range, the angle-of- L
attack vane has an added negative correction (positive error) apparently 4
due to interference from the nose of the sideslip vane. 2
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<u>Angle-of-sideslip vane</u>.- The variations of the angle-of-sideslip vane correction with angle of sideslip at various Mach numbers are given in figure 14. For Mach numbers of 1.47 and above, the influence of angle of attack on the sideslip-vane corrections is also shown. At $\alpha \approx 0^\circ$, the sideslip-vane corrections ranged from about zero to 1.5° for sideslip angles to $\pm 10^\circ$. The maximum correction for the sideslip vane at $\beta \approx 0^\circ$ and $\alpha \approx 0^\circ$ was about -0.5° and occurred at $M_\infty = 2.36$. The influence of the temperature element considerably reduced the sideslip-vane error at $M_\infty = 2.01$ reported in reference 2.

Stagnation-Temperature Element

Recordings of the stagnation-temperature element indications were taken to check its performance over the angular range covered in these tests. These measurements indicated that there were no significant variations in the measured stagnation temperature when the probe was moved through angles of attack from about -3° to 20° and/or angles of sideslip from -10° to 10°. Previous calibrations of similar elements (ref. 3) have indicated a recovery factor of about 0.99.

CONCLUDING REMARKS

Wind-tunnel calibrations of a combined pitot-static tube, vane-type flow-direction transmitter, and stagnation-temperature element over a range of Mach numbers from 0.60 to 2.87 have indicated the following:

1. There is no significant error in the measurement of total pressure resulting from angles of attack from -3° to 20° and/or angles of sideslip from -10° to 10°.

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2. At zero angles of attack and sideslip, the static-pressure error is positive in the subsonic and transonic range and becomes slightly negative in the supersonic range. The variations in static-pressure error due to angle of attack are generally less than 1 percent of the impact pressure for angles up to 15°. Because of the unsymmetrical orifice arrangement, the variations due to angle of sideslip are much larger and range from 2 to 4 percent, depending on Mach number, for a sideslip angle of 10°.

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3. The angle-of-attack vane corrections at zero angle of sideslip appear sufficiently regular to permit reasonably accurate corrections in the test range. The corrections are roughly linear with slopes ranging from about 0.5° in 15° to 2.4° in 15°, depending on the Mach number.

4. The nature of the flow field for the sideslip (rear) vane causes the magnitude and sign of the correction to vary considerably with Mach number. At zero angle of attack, the sideslip-vane corrections range from about zero to 1.5° for sideslip angles to ±10°.

5. There are no significant variations in the measured stagnation temperature due to angles of attack from about -3° to 20° and/or angles of sideslip from -10° to 10°.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., July 31, 1959.

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2. Sinclair, Archibald R., and Mace, William D.: Wind-Tunnel Calibration of a Combined Pitot-Static Tube and Vane-Type Flow-Angularity Indicator at Mach Numbers of 1.61 and 2.01. NACA TN 3808, 1956.

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6. Silsby, Norman S.: External Interference Effects of Flow Through Static-Pressure Orifices of an Airspeed Head at Several Supersonic Mach Numbers and Angles of Attack. NASA MEMO 2-13-59L, 1959.

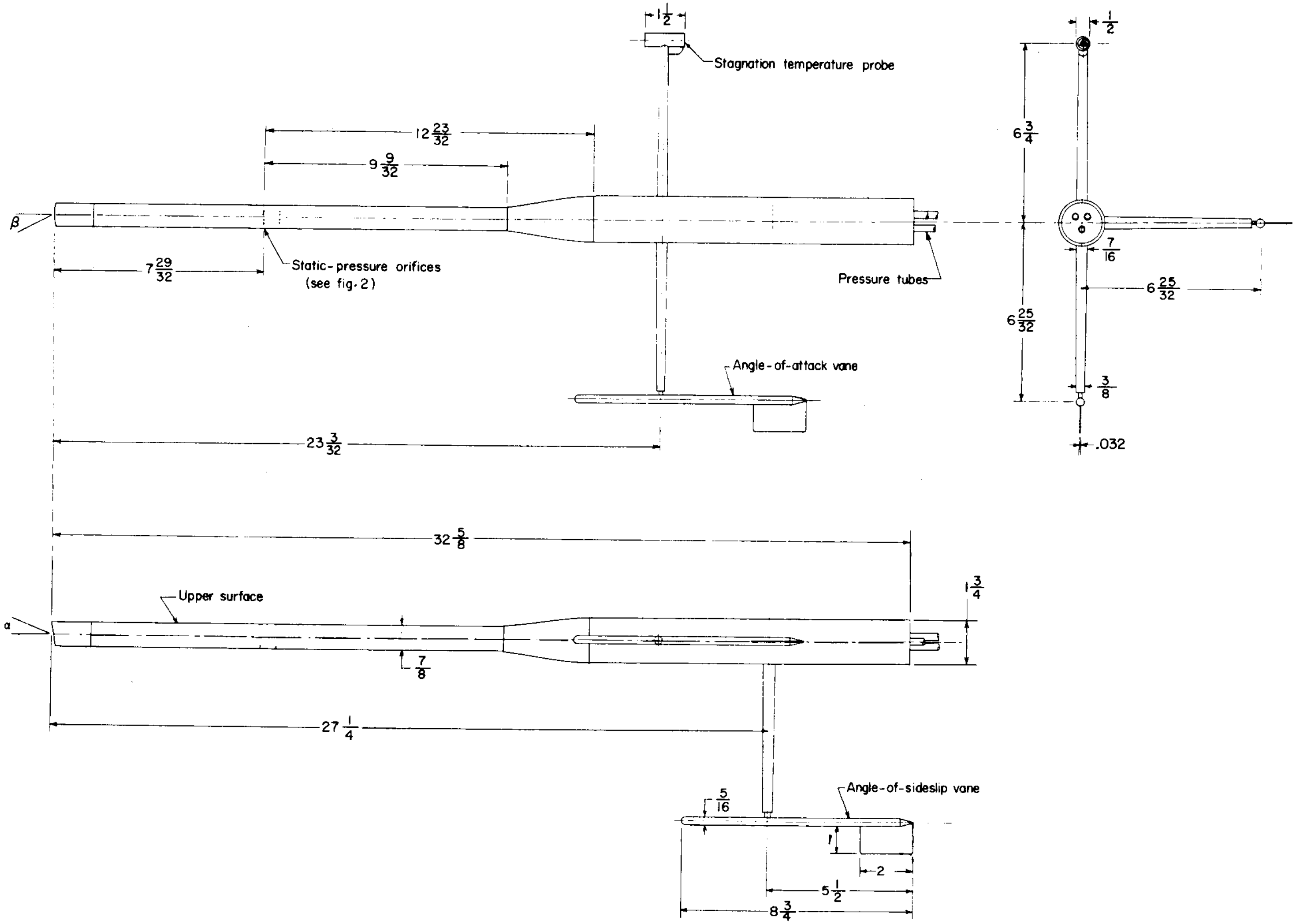

Figure 1.- Details of combined pitot-static tube vane-type flow-direction transmitter, and stagnation temperature element. All dimensions are in inches.

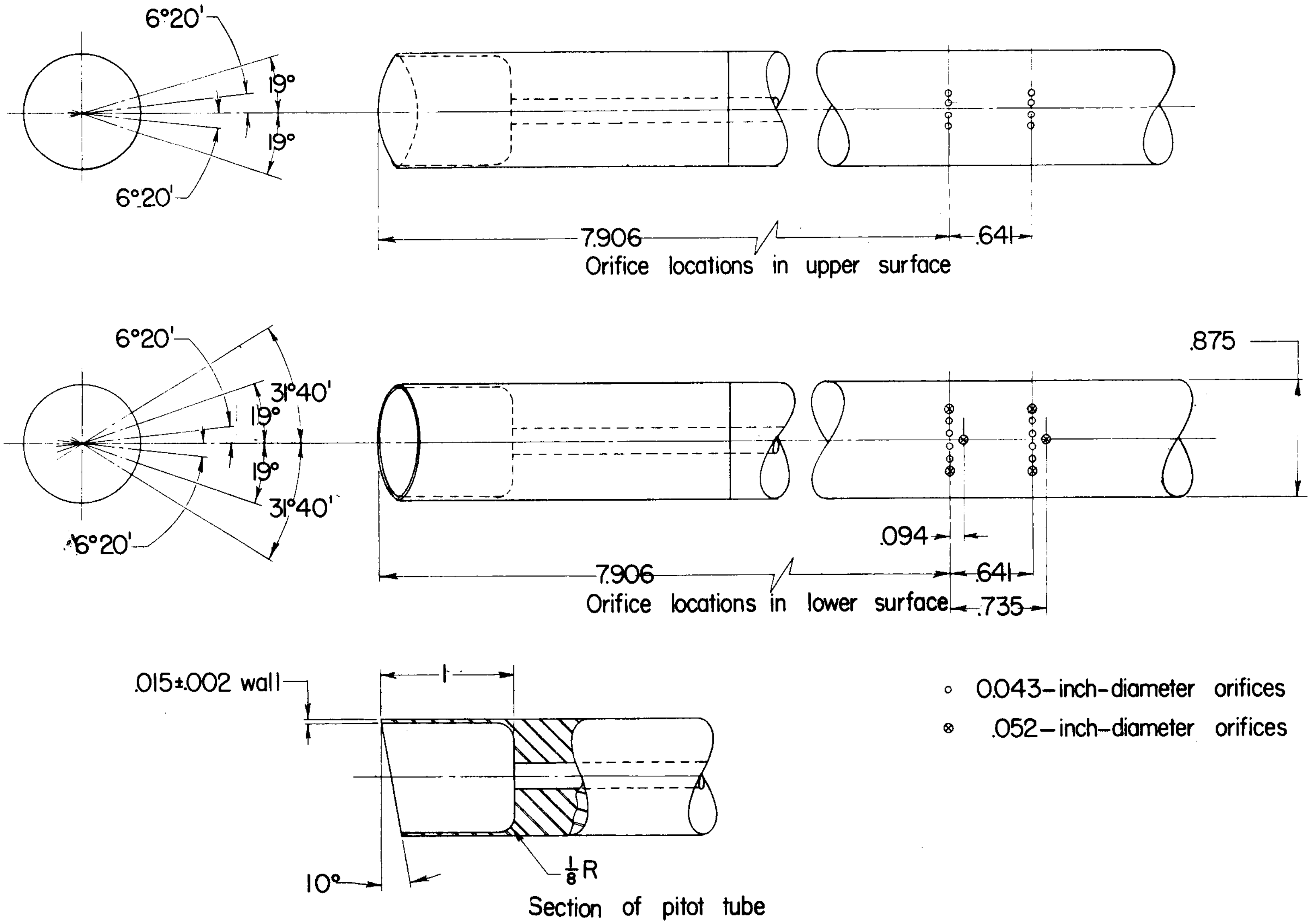

Figure 2.- Details of pitot-static tube. All dimensions are in inches.

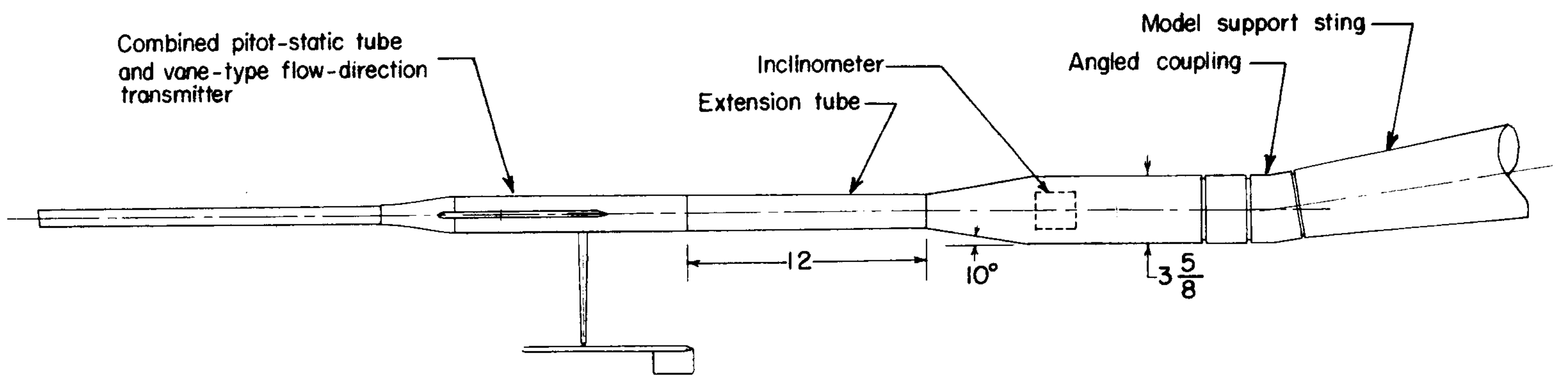

Figure 3.- Installation of instrument on adapter in Langley 8-foot transonic pressure tunnel. All linear dimensions are in inches.

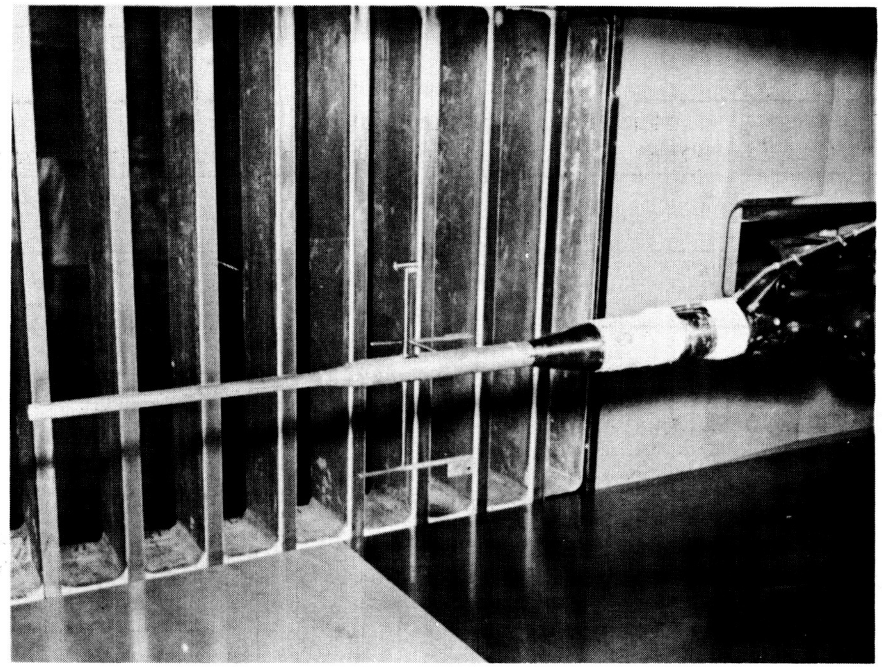

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Figure 4.- Installation of instrument on adapter in Langley Unitary Plan wind tunnel.

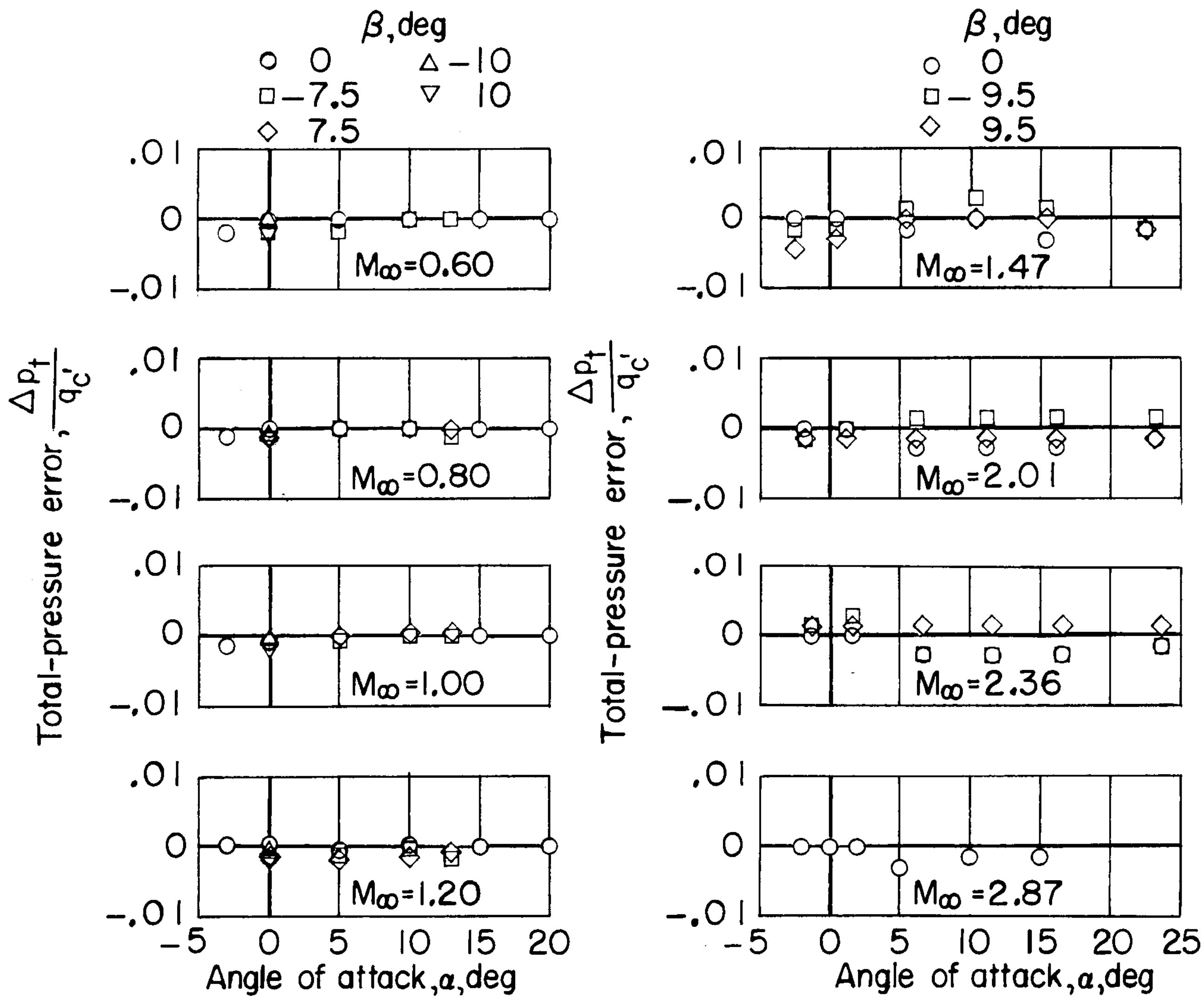

Figure 5.- Variation of total-pressure error with angle of attack at various angles of sideslip and Mach numbers.

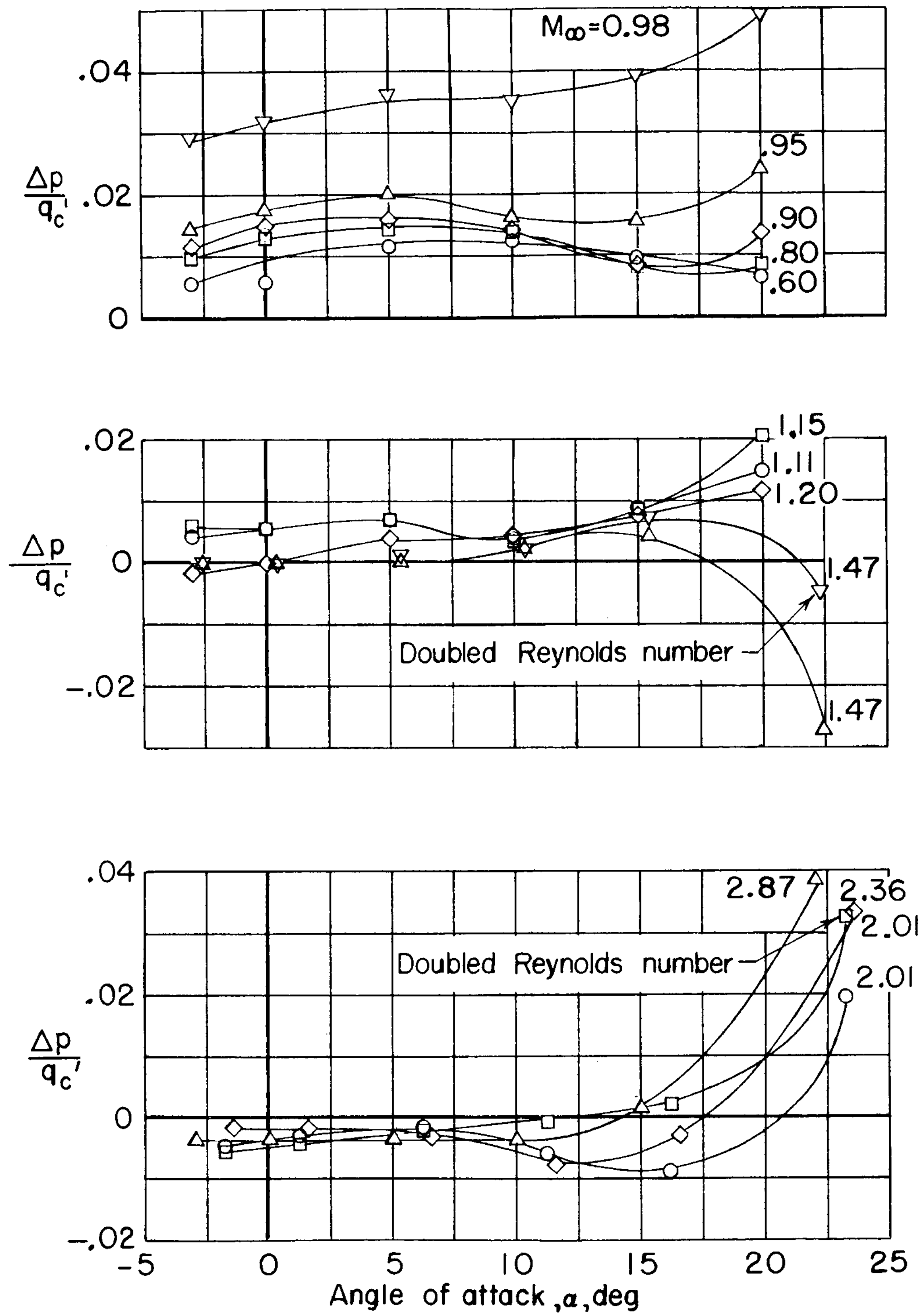

Figure 6.- Variation of static-pressure error with angle of attack. $\beta = 0^{\circ}$; forward set of orifices.

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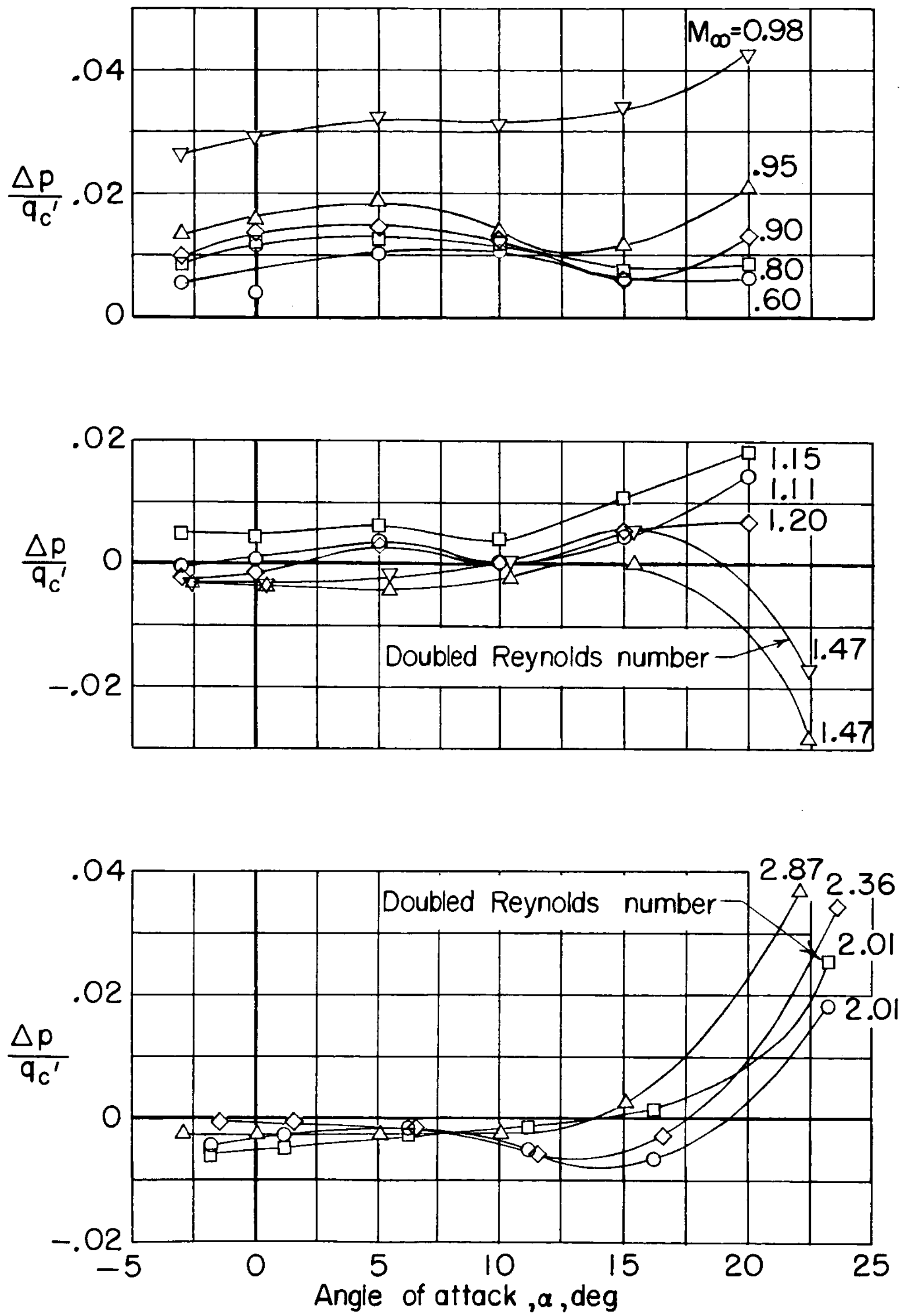
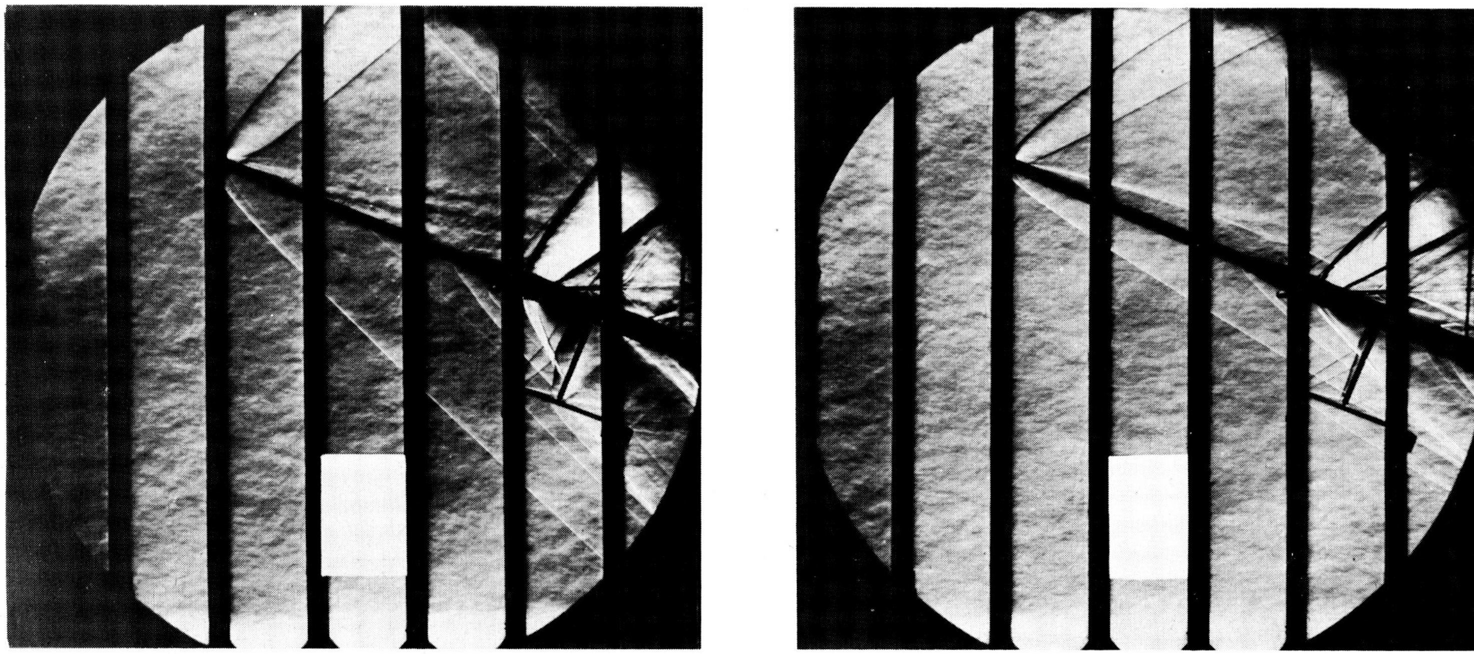


Figure 7.- Variation of static-pressure error with angle of attack. $\beta = 0^{\circ}$; rearward set of orifices.



(a) $M_\infty = 1.47$. (b) $M_\infty = 2.01$.

Figure 8.- Schlieren photographs of flow around instrument. $\alpha \approx 22^{\circ}$. L-59-5017

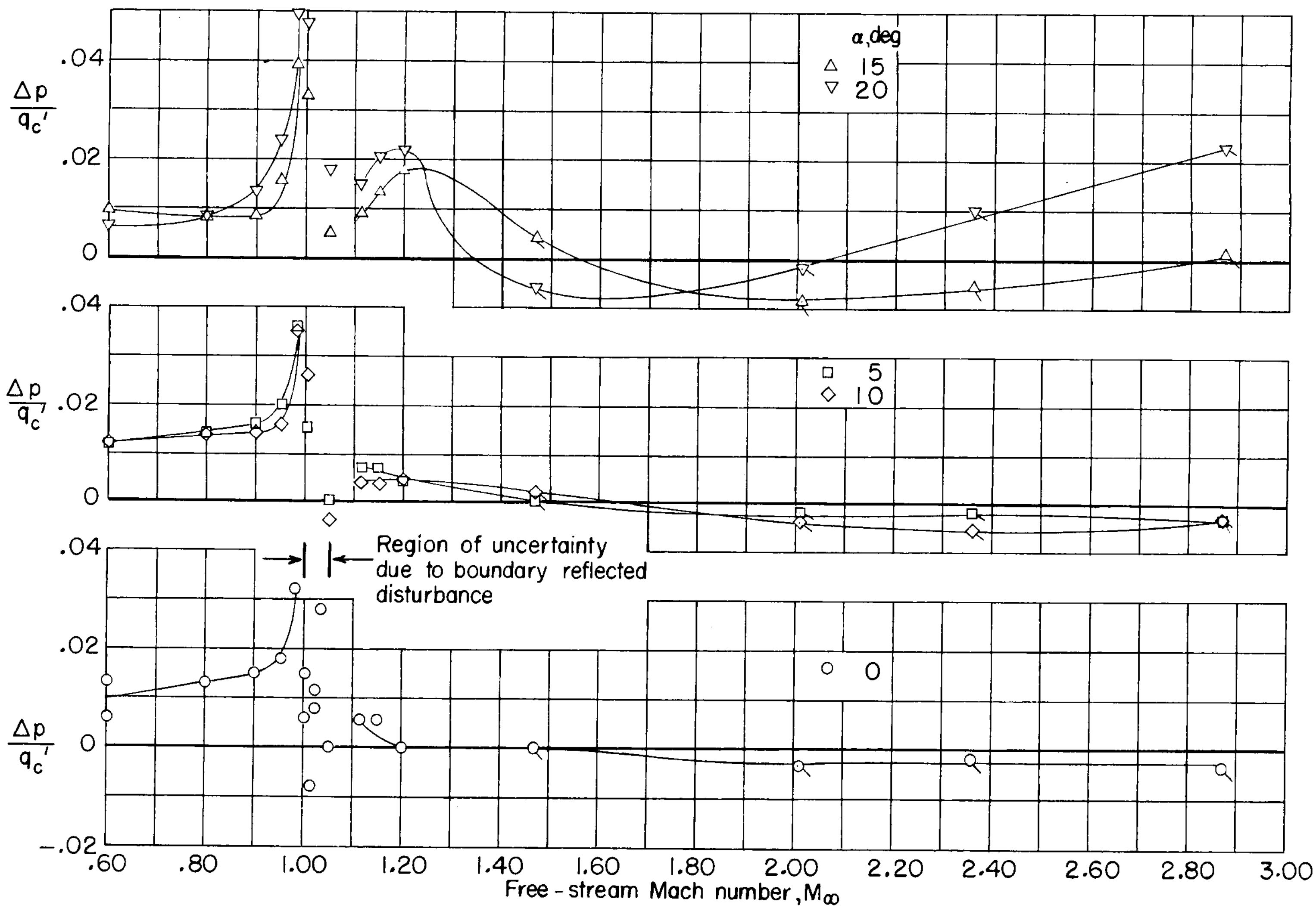


Figure 9.- Variation of static-pressure error with Mach number. $\beta = 0^{\circ}$. (Tailed test-point symbols denote data taken from fairing in fig. 6.)

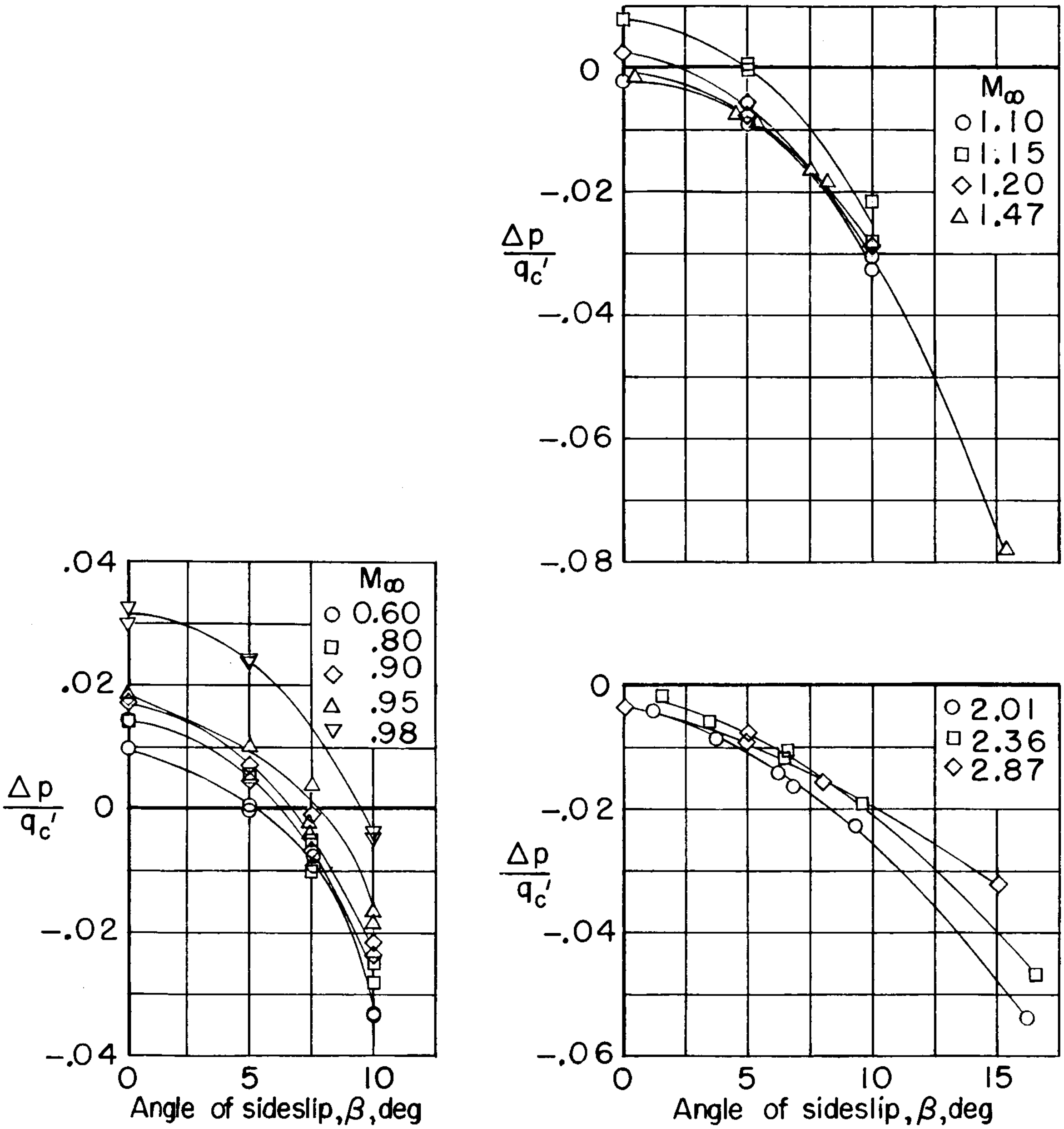


Figure 10.- Variation of static-pressure error with angle of sideslip. $\alpha = 0^{\circ}$.

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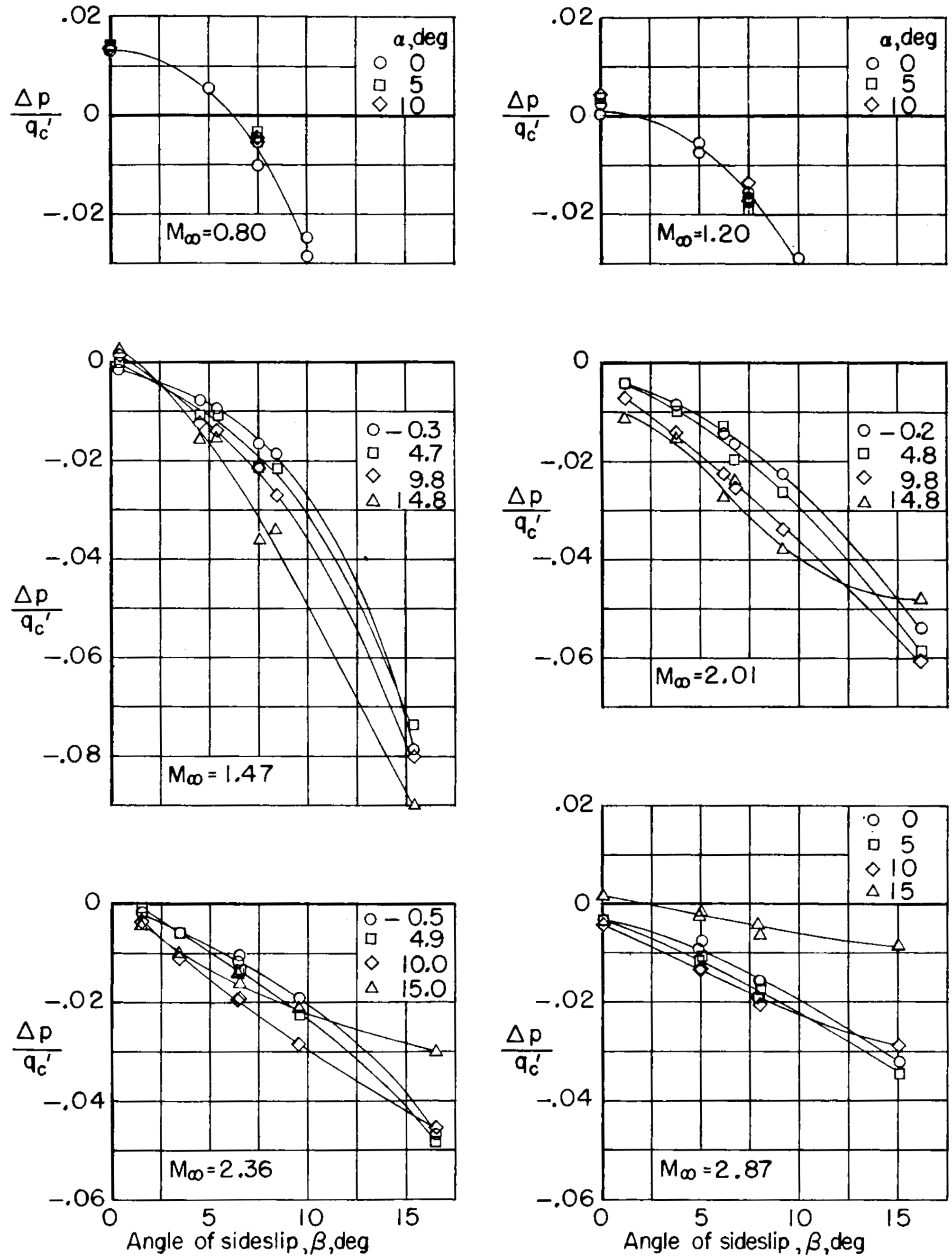
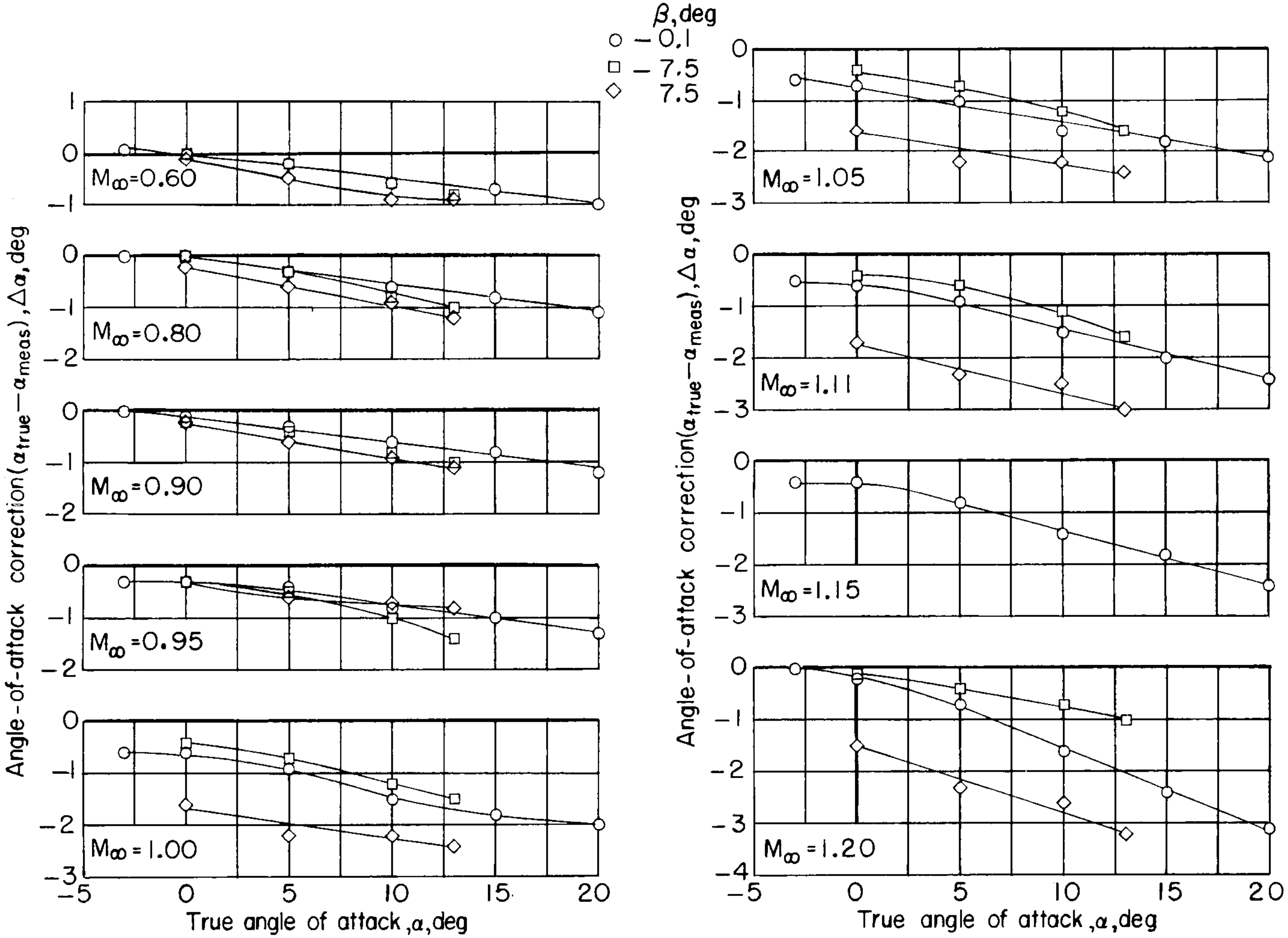
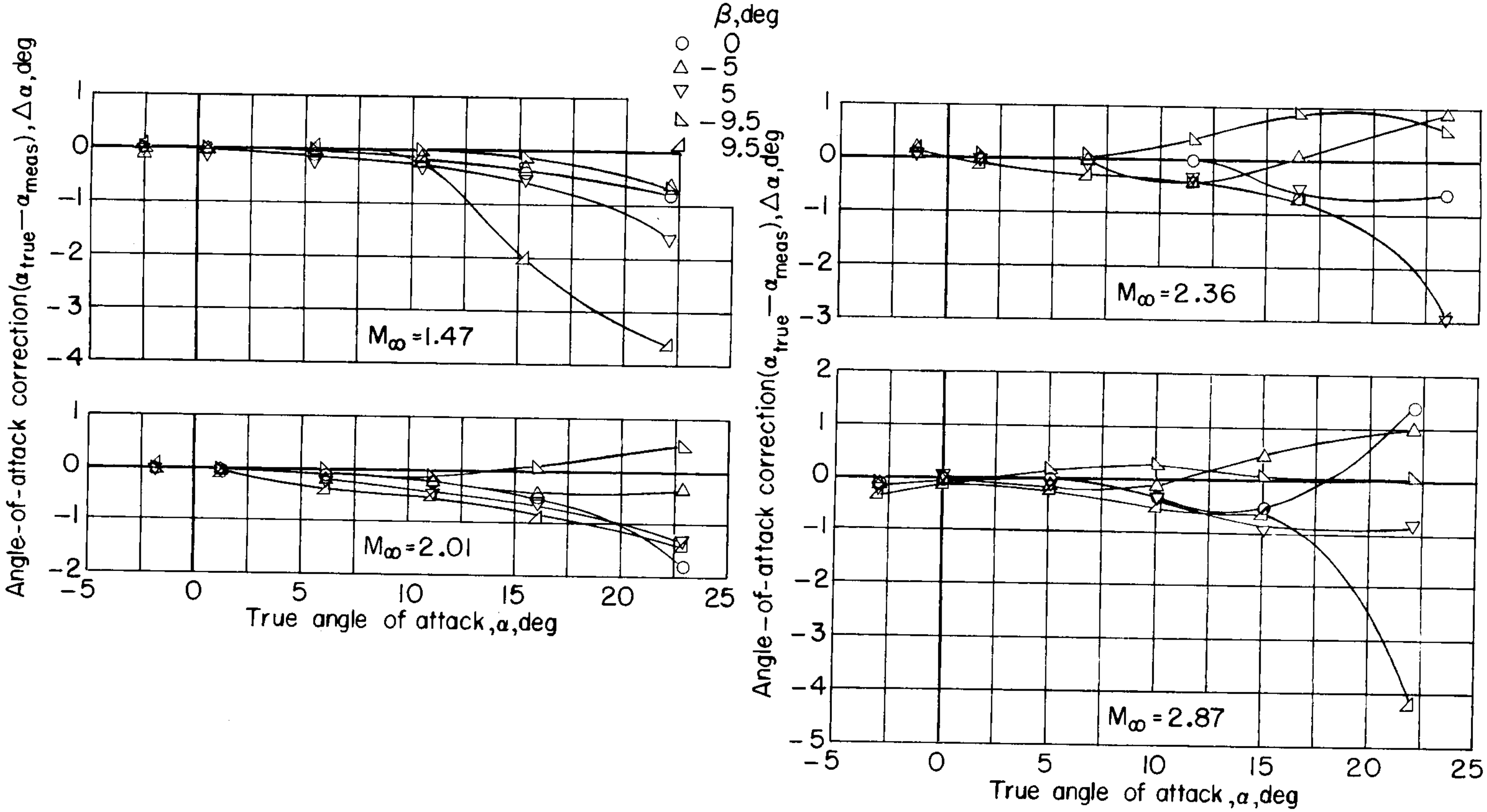


Figure 11.- Variation of static-pressure error with angle of sideslip at various angles of attack.



(a) M_∞ = 0.60 to 1.20.

Figure 12.- Variation of angle-of-attack vane correction with angle of attack.



(b) M_∞ = 1.47 to 2.87.

Figure 12.- Concluded.

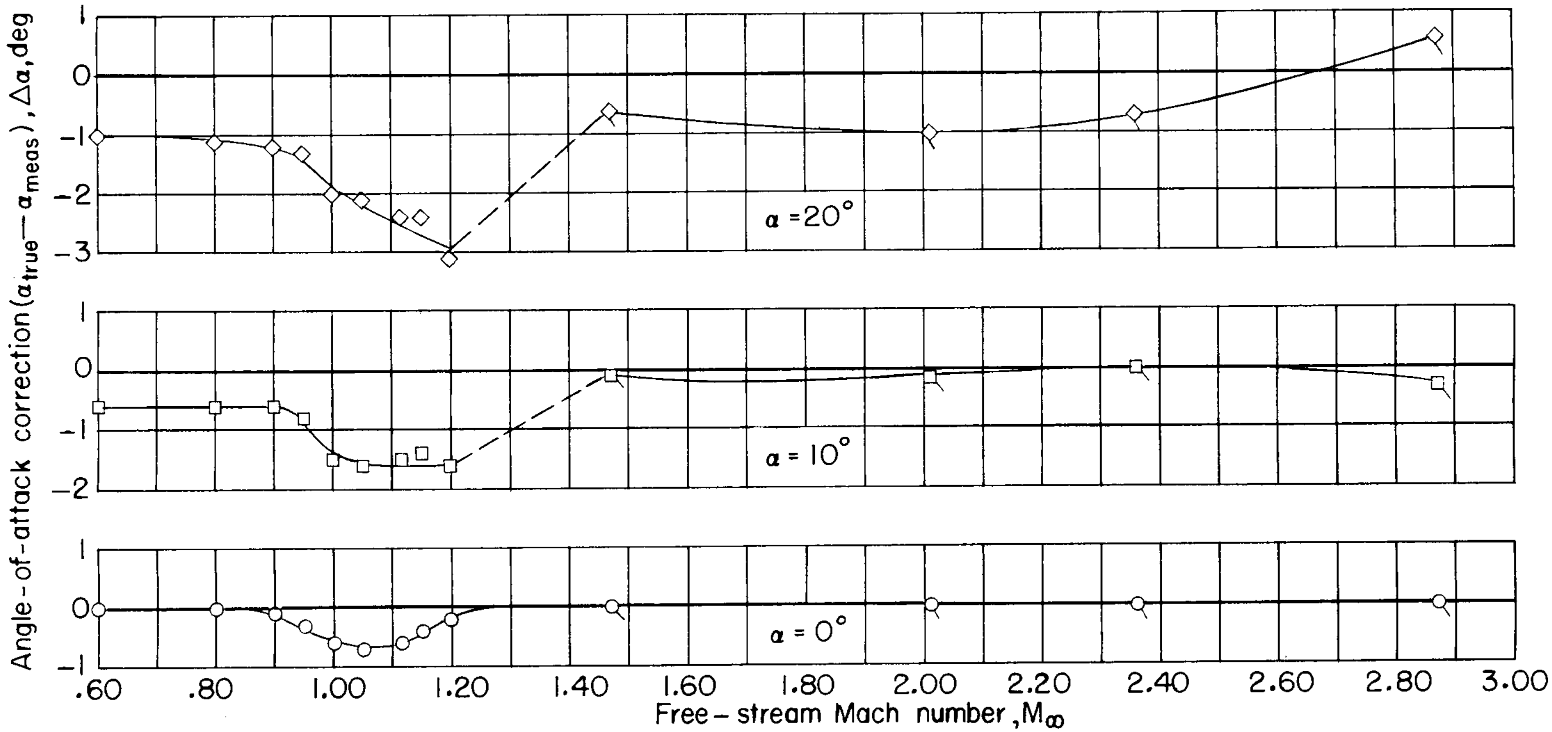
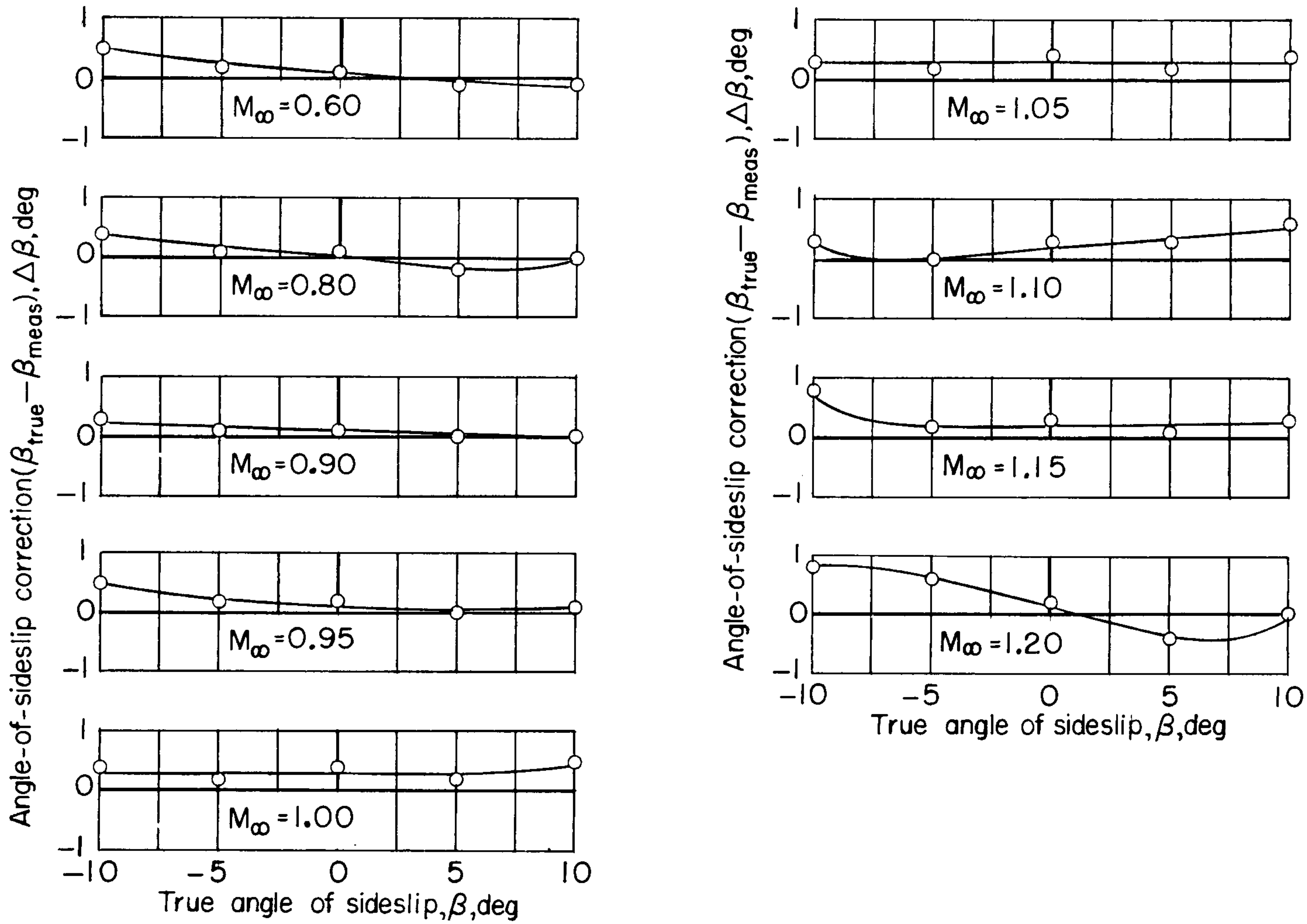
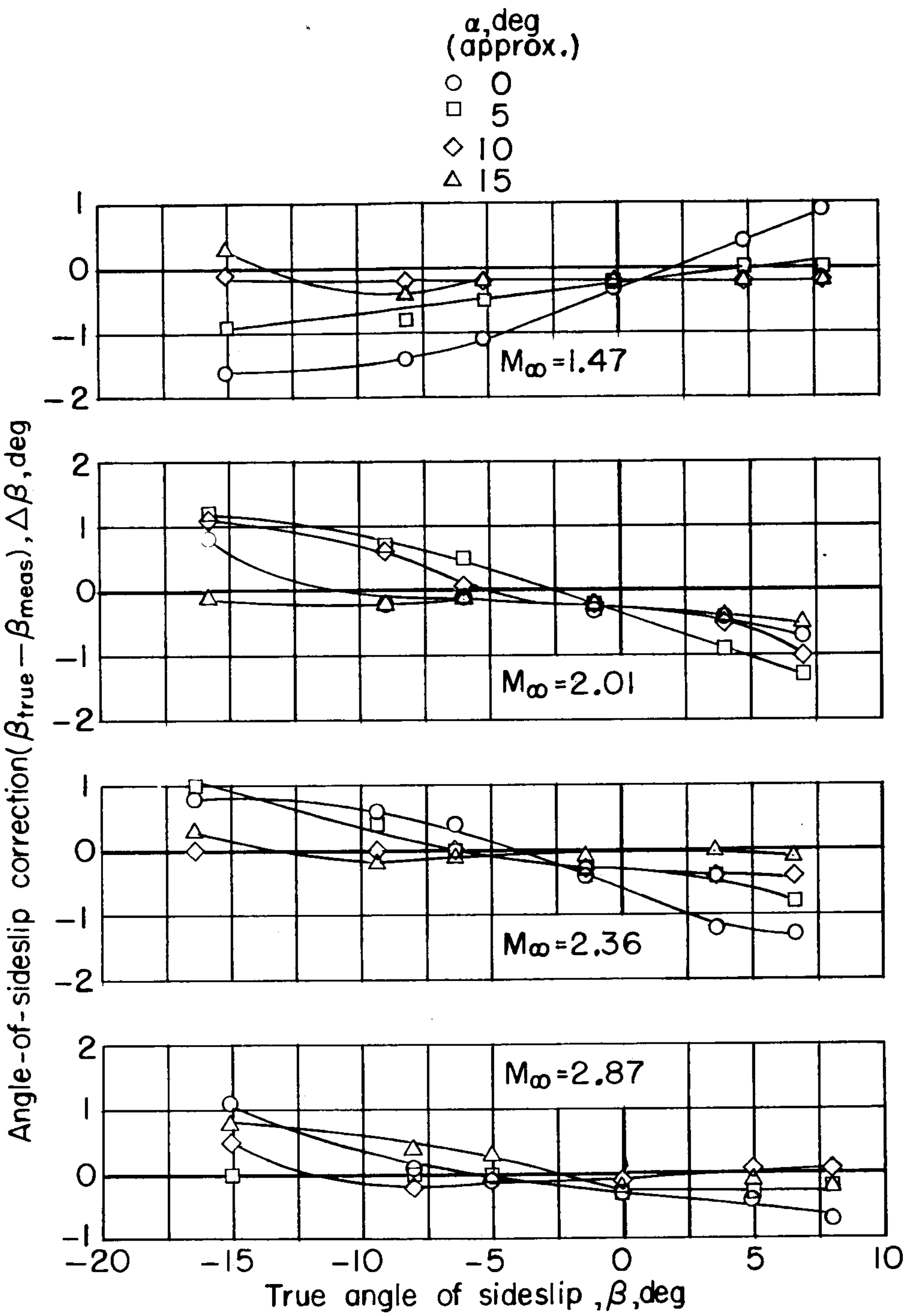


Figure 13.- Variation of angle-of-attack vane correction with Mach number. $\beta = 0^{\circ}$. (Tailed test-point symbols denote data taken from fairing in fig. 12(b).)



(a) M_∞ = 0.60 to 1.20; $\alpha \approx 0^\circ$.

Figure 14.- Variation of angle-of-sideslip vane correction with angle of sideslip.



(b) M_∞ = 1.47 to 2.87.

Figure 14.- Concluded.

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